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PRELIMINARY ABORT ΔV REQUIREMENTS FOR A COPERNICUS MISSION

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Advanced Mission Design Branch

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Approved: Jack Funk
Jack Funk, Chief
Advanced Mission Design Branch

Approved: J P Mayer
John P. Mayer, Chief
Mission Planning and Analysis Division

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PRELIMINARY ABORT ΔV REQUIREMENTS FOR A COPERNICUS MISSION

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SUMMARY

The optimum ΔV costs to abort a lunar mission for a Copernicus landing in May 1971 is presented. The abort ΔV available from the combination of an extended lunar module (ELM) or an augmented lunar module (ALM) with Apollo Block II command and service modules (CSM) hardware is also presented for the range of service module (SM) off-load conditions extending from 0 to 100 percent primary propulsion propellant through total SM jettison. These results are cross-plotted to determine the propellant off-load requirements for accomplishing the abort maneuvers considered.

The abort maneuvers considered in this study were initiated at an altitude of 80 n. mi. at pericynthion for nonfree-return translunar trajectories having flight times of 110, 95, 90, 85, and 80 hours. Abort maneuvers initiated from the 95-hour translunar trajectory at times 1 and 2 hours after pericynthion passage were also investigated. The analysis was terminated at transearth injection (TEI) on the lunar sphere of influence (LSOI). This target point was corrected in longitude from the nominal Copernicus mission of reference to account for the portion of the total abort flight time spent within the LSOI at the rate of 13.18° per day. Only this portion of the total abort trajectory was optimized, as the optimization program used in this study is limited to operate solely within the influence of a single attracting body.

The results of the optimization study were plotted as a function of time to interpolate for the local minimum ΔV costs and the time associated with these values. The total abort flight time required to reach the reentry point on the Earth was determined by adding 89.1 hours to the interpolated values.

For the range of abort maneuvers investigated for which only the descent propulsion system (DPS) is used, service propulsion system (SPS) propellant off-load requirements of 52.0 to 67.6 percent were required for a vehicle equipped with the ELM, and 21.0 to 41.4 percent if equipped with the ALM.

The propellant off-load requirements were found to increase with increase in translunar flight time. The interpolated local minimum ΔV costs established for aborting this mission from the above stated trajectories are 2750, 2637, 2600, and 2570, and 2535 fps, respectively. The stay times within LSOI associated with these ΔV values averaged 2.75 ± 0.05 days.

INTRODUCTION

The purpose of this effort was to determine both the required and available ΔV magnitudes for aborting a lunar mission for a Copernicus landing in May 1971. This mission is described in detail in reference 1, and is typical of the trajectory planning for advanced lunar missions discussed in references 2 and 3. These missions require (1) nonfree-return translunar trajectories, (2) Apollo Block II CSM hardware, (3) either an ELM or an ALM (see refs. 4, 5, and 6), and (4) thrust abort capability. For example, SPS failure at lunar orbit insertion (LOI) requires that the spacecraft possess additional and separate propulsion systems to take the vehicle through TEI at the LSOI. In this study, LOI was considered to take place at pericynthion; but, since abort conditions are simulated, the initial abort thrust was assumed to take place immediately at LOI. However, preparatory pre-abort vehicle orientation maneuvers will be required before abort maneuvers can be initiated.

This study presents (1) the optimum ΔV costs to abort a Copernicus mission from nonfree-return translunar trajectories having a range of flight times of 110 to 80 hours, and (2) ΔV available from the propulsion systems (descent and ascent) of an ELM or an ALM for a range of SM off-load conditions extending from 0 to 100 percent primary propulsion propellant through total SM jettison. The translunar trajectory having the 110-hour flight time was used as the basis for the trajectory computations. The configuration weights were obtained from references 4, 5, and 6.

Two separate digital computer programs were used in this study. The Apollo Trajectory Design Program (ATDP), described in reference 7, was used to determine the departure and target state vectors for the abort maneuvers. An optimal n-impulse trajectory program, described in detail in reference 8, was used to determine the optimal ΔV costs for the abort maneuvers considered.

The optimum ΔV costs computed in this study are subject to the limitations of the optimization program. As a result, only that portion of the abort trajectory lying within the LSOI was optimized, as the program does not have the capability of traversing an SOI. Further, the optimization program assumes infinite vehicle thrust capability, which is vastly different from the capability of the vehicles considered in this study where thrust-to-weight ratio is always much less than unity. As a result, it is conceivable that the optimum ΔV costs computed in this study are appreciably different from the results of a more realistic integrated trajectory. However, this discrepancy does not detract significantly from this study, as the results show that the abort cannot be successfully accomplished unless the vehicle is reduced in weight by a very wide margin.

No attempt was made in this analysis to determine the dynamic stability of the vehicle in regard to propellant off-loading. However, any future work on this subject should be intimately concerned with the problem of thrust misalignment resulting from propellant off-loading and a thrusting mode of operation which is the direct opposite of the design mode for the Apollo mission. If the thrust misalignment problems cannot be contained within the control bounds of the stabilization and control systems, it will not be possible to supply Copernicus-type missions with thrust abort capability from the ELM or ALM propulsion systems.

SYMBOLS

g	acceleration of gravity for the Earth, 32.17 ft/sec^2
I_{SP}	specific impulse for the propellant combination of each lunar module propulsion system, 300 sec
M_o	total weight of vehicle at the time of the initial abort impulse, lb
M_D	total weight of descent stage jettisoned, lb
M_1	total usable weight of descent propulsion system propellant, lb
M_2	total usable weight of ascent propulsion system propellant, lb
ΔV_A	ΔV capability of the ascent stage, fps
ΔV_D	ΔV capability of the descent stage, fps
ϕ	longitude of the target point at the completion of the abort maneuver, deg

ϕ_o longitude of the target point at TEI from a nominal Copernicus mission, 73.511239°

$\dot{\phi}$ angular velocity of the Moon about Earth, 13.18 deg/day

t_f time of flight of the abort trajectory lying within the LSOI, days

ANALYSIS

The abort maneuvers considered in this study were initiated from an altitude of 80 n. mi. at pericynthion for translunar trajectories having flight times of 110, 95, 90, 85, and 80 hours. Abort maneuvers initiated from the 95-hour translunar trajectory at times 1 and 2 hours after pericynthion passage were also investigated.

The state vector at the LSOI and TEI from the nominal Copernicus mission - described in detail in reference 8 - was taken as the target for all abort maneuvers considered. The longitude of the target was corrected for the arc traversed by the Moon about the Earth during the portion of the total flight time of the abort spent within the LSOI according to the relationship,

$$\phi = \phi_o + \dot{\phi} t_f$$

The abort ΔV available from the ELM and ALM was computed according to the relations

$$\Delta V_D = I_{sp}\, g \ln \left(\frac{M_o}{M_o - M_1} \right)$$

and

$$\Delta V_A = I_{sp}\, g \ln \left(\frac{M_o - \left(M_D + M_1\right)}{M_o - \left(M_D + M_1 + M_2\right)} \right)$$

The configuration weights for these calculations are given in table I. The results of these calculations are plotted in figures 2 and 3.

For the purpose of the optimization study, the portion of the abort trajectory lying within the LSOI was considered to require a flight time of from 1 to 5 days. Only this portion of the total abort trajectory was optimized, as the present version of the optimization program can only optimize the ΔV cost for trajectories that lie entirely within the SOI of an attracting body. As a result, the program was supplied with a fixed flight time which was only a portion of the total return time of the abort. However, repeated computer runs were made - increasing the flight time in each case in half-day increments - in an attempt to establish the local minimum for the ΔV cost curves plotted as functions of time (fig. 4). Where it was considered applicable, shorter time increments were also considered. The flight time for the portion of the trajectory spent within the SOI must be increased by 89.1 hours to obtain the total return time of the abort.

For the remainder of this discussion, the terms "abort flight time" and "abort trajectory" will be considered to mean the time and portion of the abort trajectory lying within the LSOI.

RESULTS AND DISCUSSION

The results of the capability study are presented in figures 1, 2, and 3. The results of the optimization study are presented in figure 4. A representative trajectory associated with the results shown in figure 4 is presented in figure 5(a).

The results of the optimization study show that the abort maneuvers follow two different trajectory patterns. The trajectory shown in figure 5(b) is typical of the abort trajectories which had relatively short flight times. This trajectory was obtained for an abort maneuver initiated from the 95-hour translunar trajectory and had a flight time of 1.5 days. The cost of abort maneuvers requiring trajectories of this type were considered too high to be of interest and were not included in the results presented in figure 4.

The trajectory shown in figure 5(a) is typical of abort trajectories which had relatively long flight times. All trajectories found to follow this pattern of sub-arcs had ΔV costs ranging from 2500 to 3000 fps from a total of four impulses. For these cases, the optimum trajectory required that the initial impulse be applied - not at pericynthion - but at a time approximately 13 to 14 minutes after pericynthion passage.

Cost savings of the order of 50 to 70 fps were realized through this preliminary coast as opposed to the same maneuver initiated at pericynthion. This saving is a significant factor, for it allows a period of time in which to prepare the spacecraft for the abort maneuver without penalty of cost. It may be seen from figure 4 that the penalty for postponing the abort maneuver by as much as 1 hour after pericynthion passage increases the cost of the abort by approximately 120 fps when aborting from the 95-hour trajectory. This delay also imposes a penalty on the time of flight of the abort. For the case cited, the penalty is approximately 0.25 day. If the abort is delayed by as much as 2 hours, the ΔV penalty increases to 333 fps and the time increases nearly 0.75 day. The actual time of flight of the last maneuver is approximately 3.5 days as opposed to 2.75 days for the pericynthion case.

The trajectories shown in figures 5(a) and (b) are both optimum trajectories. However, they are vastly different in cost as a result of differences in their flight times and subsequent target longitudes. An effort was made to determine the flight time associated with the lowest cost for each abort maneuver considered. However, it was found that the ΔV cost curves when plotted as a function of time show a marked discontinuity at the time where the trajectory takes a smooth elliptic path around the attracting body in place of the sharp transition to a hyperbolic sub-arc in front and away from the center of attraction. This point was difficult to establish with any degree of certainty. As a result, the ΔV cost curves presented in figure 4 have poorly defined local minimums. This was especially true of abort maneuvers initiated from translunar trajectories having the shorter LSOI stay times.

The interpolated local minimums obtained in figure 4 were cross-plotted on figures 2 and 3. It may be seen from the resulting plots that it is not possible to successfully accomplish any abort maneuver with the DPS alone unless the SPS is appreciably off-loaded. For the range of abort maneuvers investigated in which only the DPS is used, SPS propellant off-load requirements of 52.0 to 67.4 percent were required for a vehicle equipped with the ELM and 21.0 to 41.4 percent if equipped with the ALM. Abort off-load requirements increased with increase in the translunar flight time of the initiating trajectory. A free-return translunar trajectory requires a flight time of approximately 72 hours. However, it may be seen from figure 6 that the payload capability of a mission can be increased by approximately 1000 lb if the 80-hour nonfree-return translunar trajectory is used instead of the free-return trajectory.

It is conceivable that SPS propellant off-loading must be carried out in the same ratio by weight - oxidizer to fuel - as the operating mixture ratio of the SM engine. If this is not done, the resulting shifts in the center of mass will probably exceed the control limits of the stabilization system. For this reason it was not considered advisable to include the ascent stage in the capability of these systems for abort considerations. However, its contribution was included in figures 1, 2, and 3 to show the additional capability required.

CONCLUSIONS

Copernicus missions employing nonfree-return translunar trajectories require thrust abort capability. However, the thrust abort capability available from the ELM or the ALM is insufficient for a successful abort unless the SPS is off-loaded of a large amount of propellant. The propellant off-load requirements increase with increase in translunar flight time above the free-return flight time. Thrust abort capability becomes feasible only if it is possible to use both the descent and ascent propulsion systems to accomplish the abort maneuvers.

The optimum ΔV cost for the abort is realized by delaying the initial thrust by a few minutes after pericynthion passage. Advantage can be taken of this time to perform required pre-abort vehicle orientation maneuvers. However, delaying the initiation of the abort maneuver by approximately 1 hour will increase the cost of the abort by approximately 100 fps.

The capability of the DPS to accomplish the abort maneuver investigated is predicated on having the additional capability of off-loading a large percentage of SPS propellant. Even with this capability, an off-loading operation would require an appreciable amount of time. This is primarily due to the hypergolic nature of the propellant combination of the SPS which constitutes a potential hazard to the spacecraft. However, as the most promising abort maneuvers require approximately 2.75 days, an appreciable amount of propellant could be off-loaded during the time the vehicle is coasting between impulses. The off-loading, however, must be carried out in such a manner as to maintain vehicle center-of-mass variations within the control bounds of the stabilization system. Due to the difficulties of measuring propellant quantities in a zero-gravity environment, it is recommended that if propellant off-loading is to be performed, the SPS should be completely emptied of propellant prior to the initiation of the abort maneuver. If this can be accomplished within 2 hours, and excess ΔV of approximately 500 fps will result if the ELM is used and approximately 1000 fps of the ALM is used. If the off-loading operation cannot be accomplished, it will be necessary to consider a different solution which will permit an abort within the limitations of the available propulsion systems.

TABLE I.- CONFIGURATION WEIGHT SUMMARY

	Extended LM, lb	Augmented LM, lb
Descent stage		
Inert stage	5 398	7 510
DPS propellant	17 510	20 798
Total	22 908	28 308
Ascent stage		
Inert stage	4 690	4 690
RCS propellant (usable)	575	575
APS propellant (usable)	4 827	4 827
Total	10 092	10 092
Total LM weight	33 000	38 400
Service module propellant weight	37 325	34 100
Total configuration weight	95 225	97 400

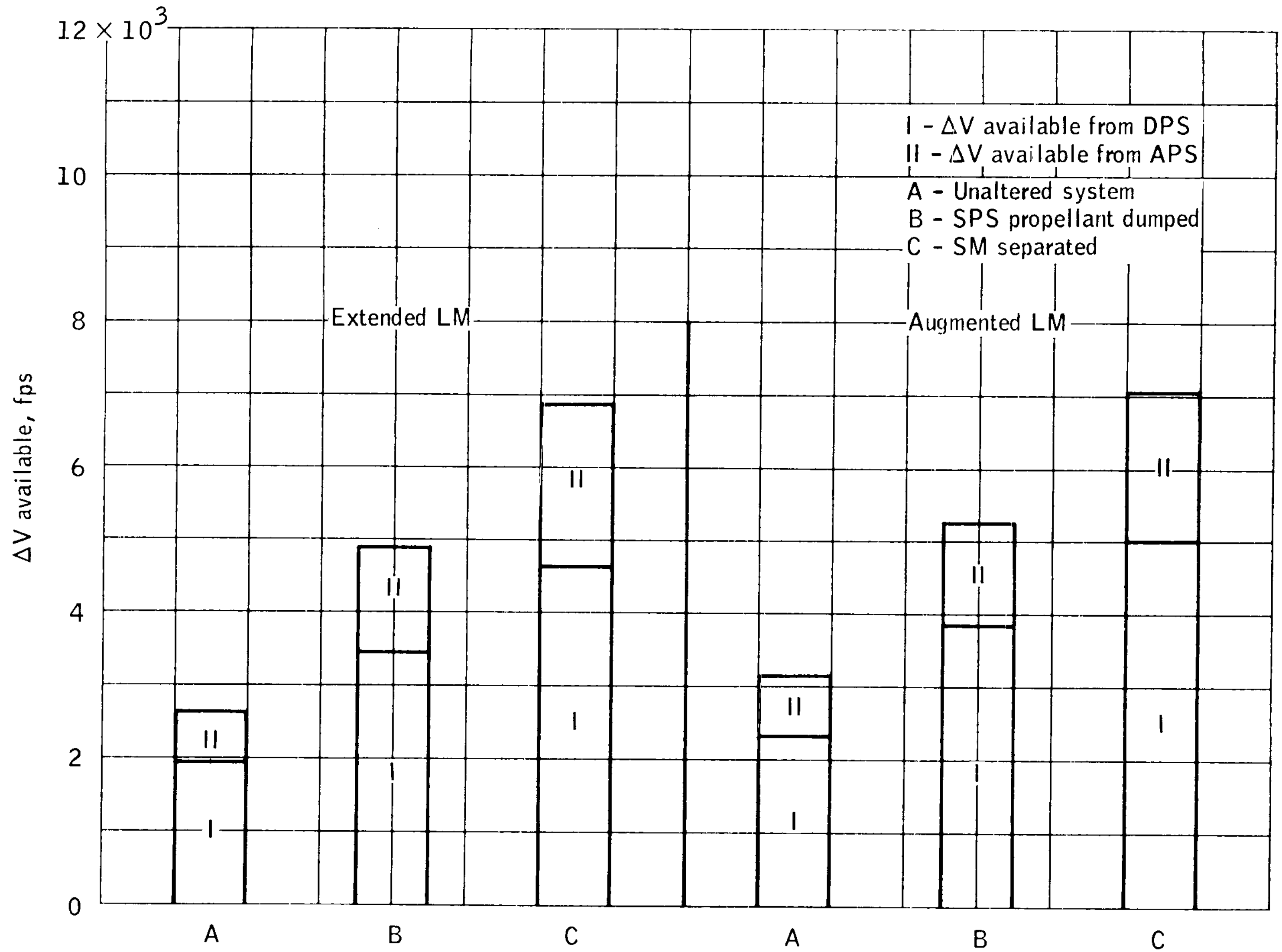


Figure 1.- Total ΔV available for abort as the SPS is jettisoned.

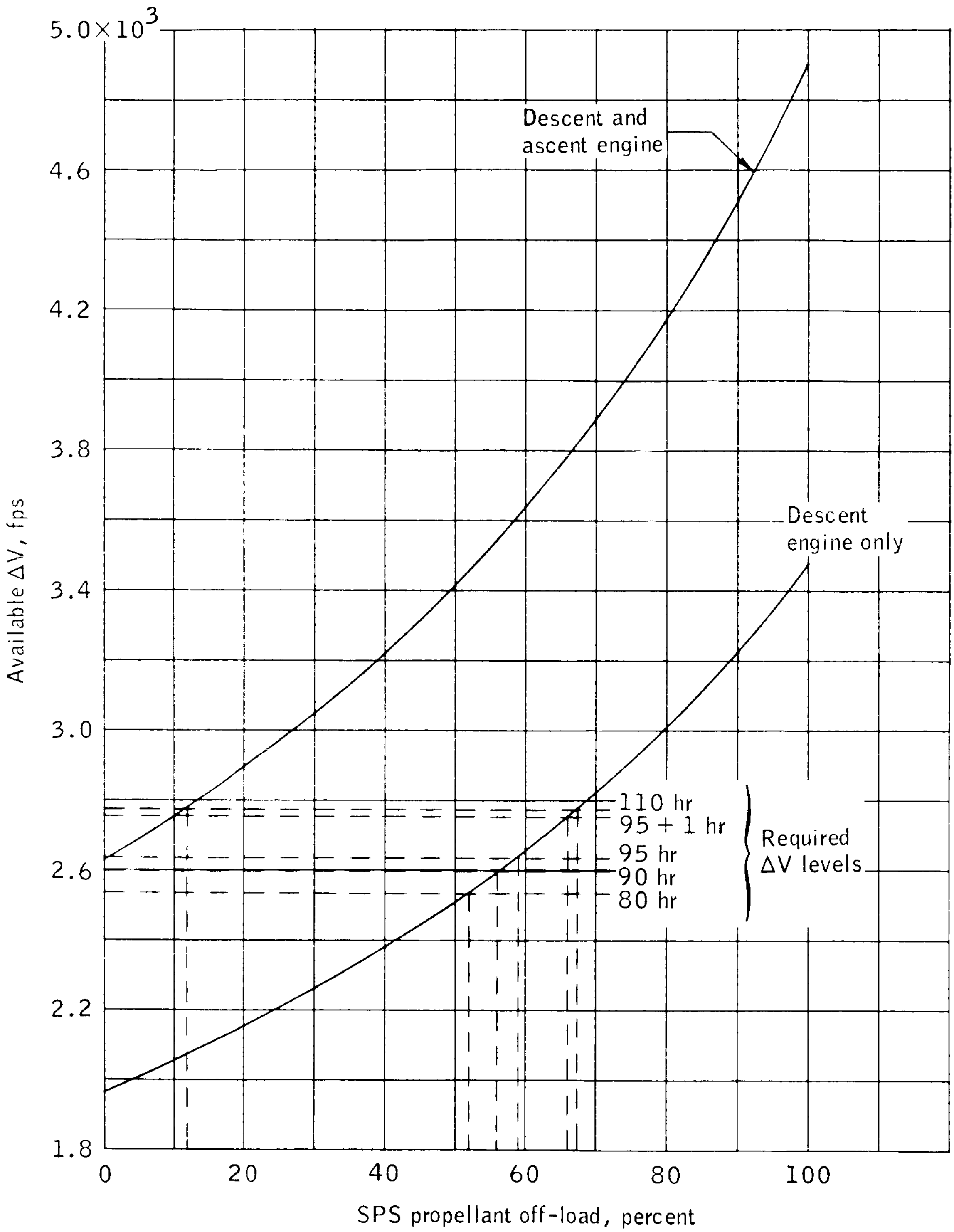


Figure 2.- Available and required ΔV for various off-load conditions of SPS propellant using the extended LM configuration.

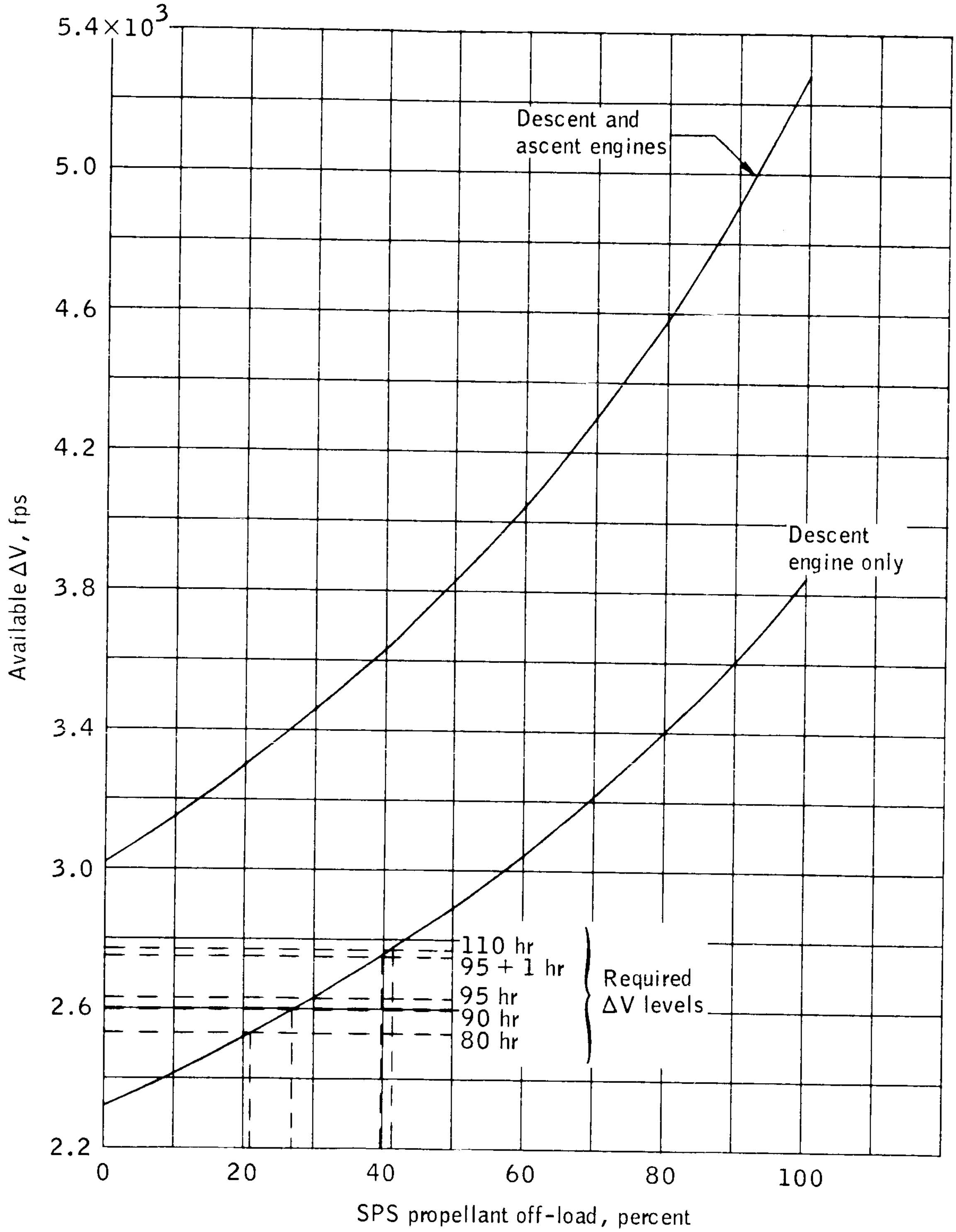


Figure 3.- Available and required ΔV for various off-load conditions of SPS propellant using the augment LM configuration.

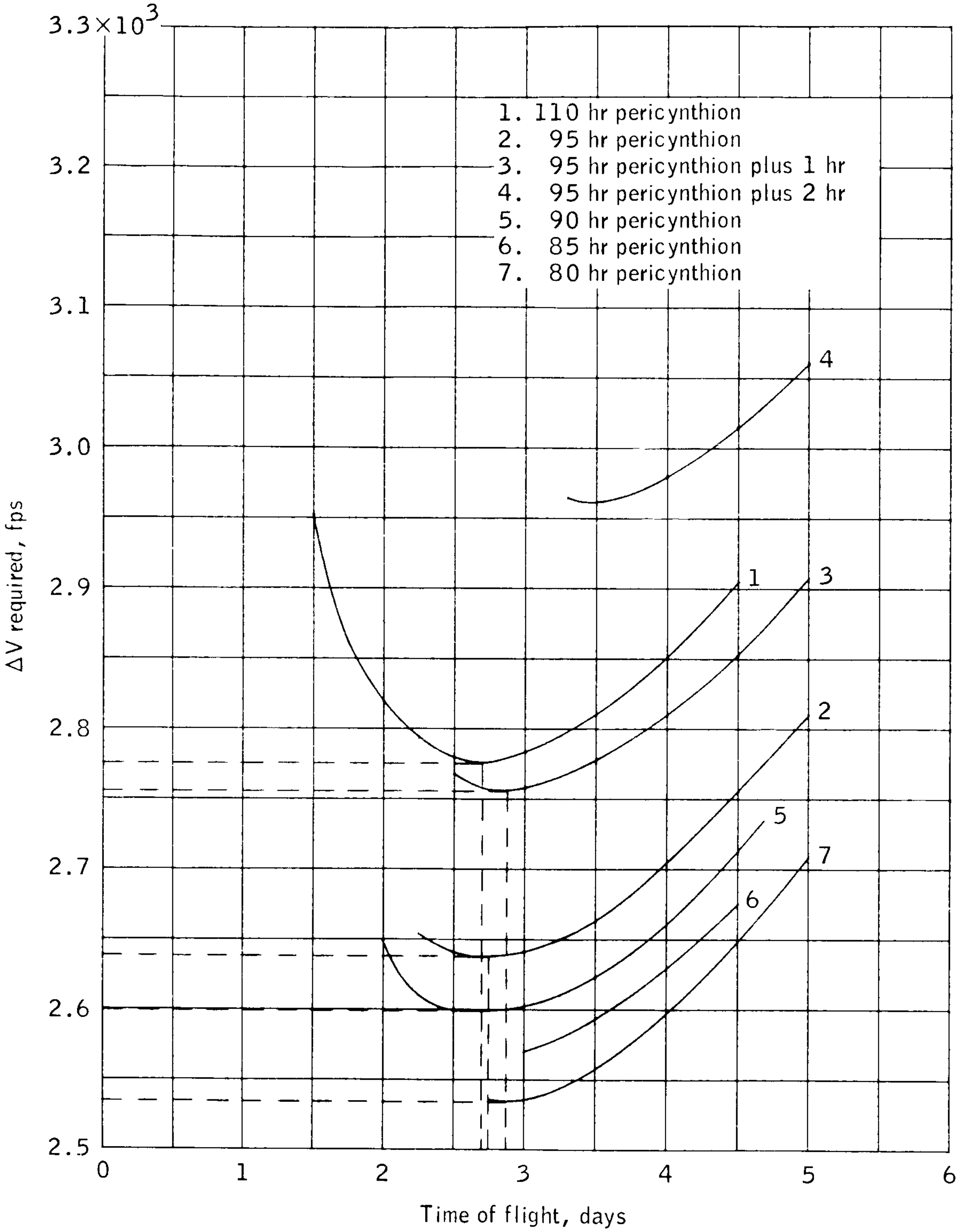
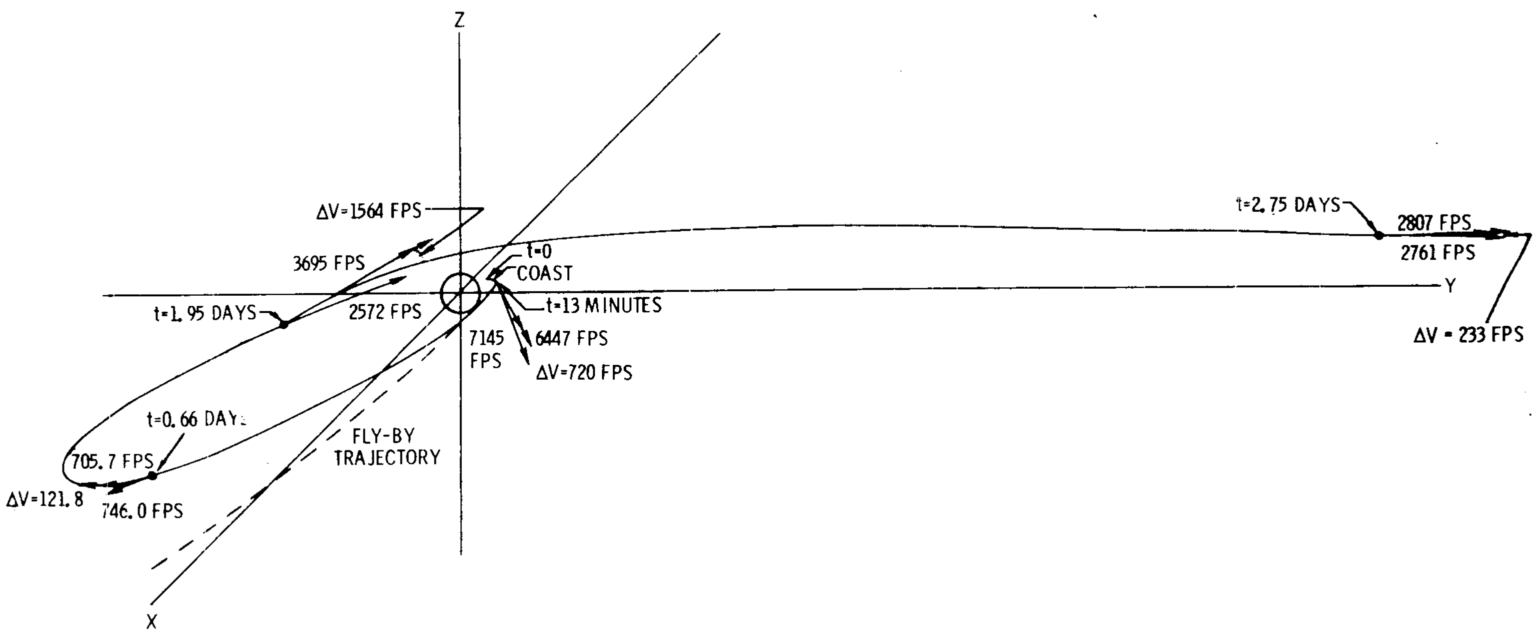
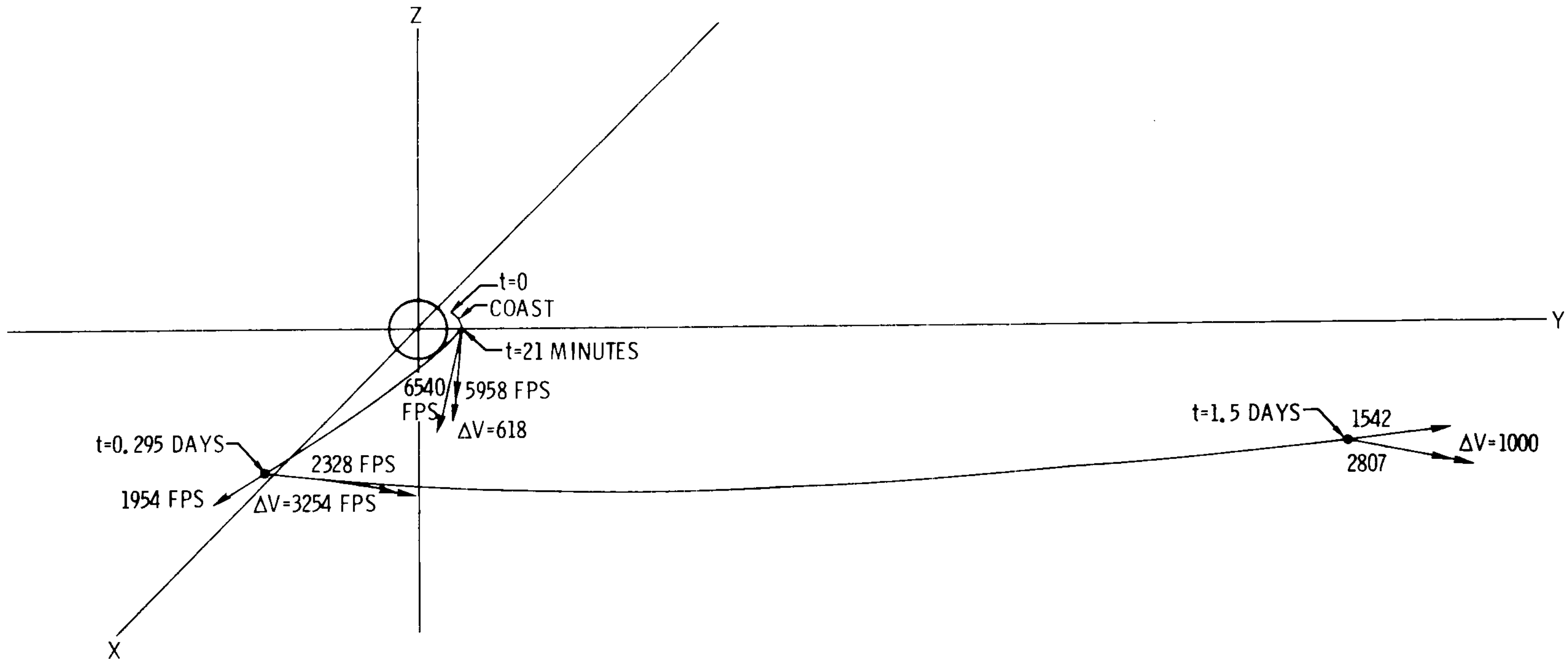


Figure 4.- Abort ΔV requirements from various translunar trajectories as a function of abort maneuver flight time.



(a) Total flight time of abort maneuver = 2.75 days; total abort ΔV = 2638 fps.

Figure 5. - Optimum trajectory pattern for an abort maneuver from a 95-hour translunar trajectory showing time, position and magnitude of impulses.



(b) Total flight time of abort maneuver = 1.5 days; total abort ΔV = 5763 fps.

Figure 5.- Optimum trajectory pattern for an abort maneuver from a 95-hour translunar trajectory showing time, position and magnitude of impulses.

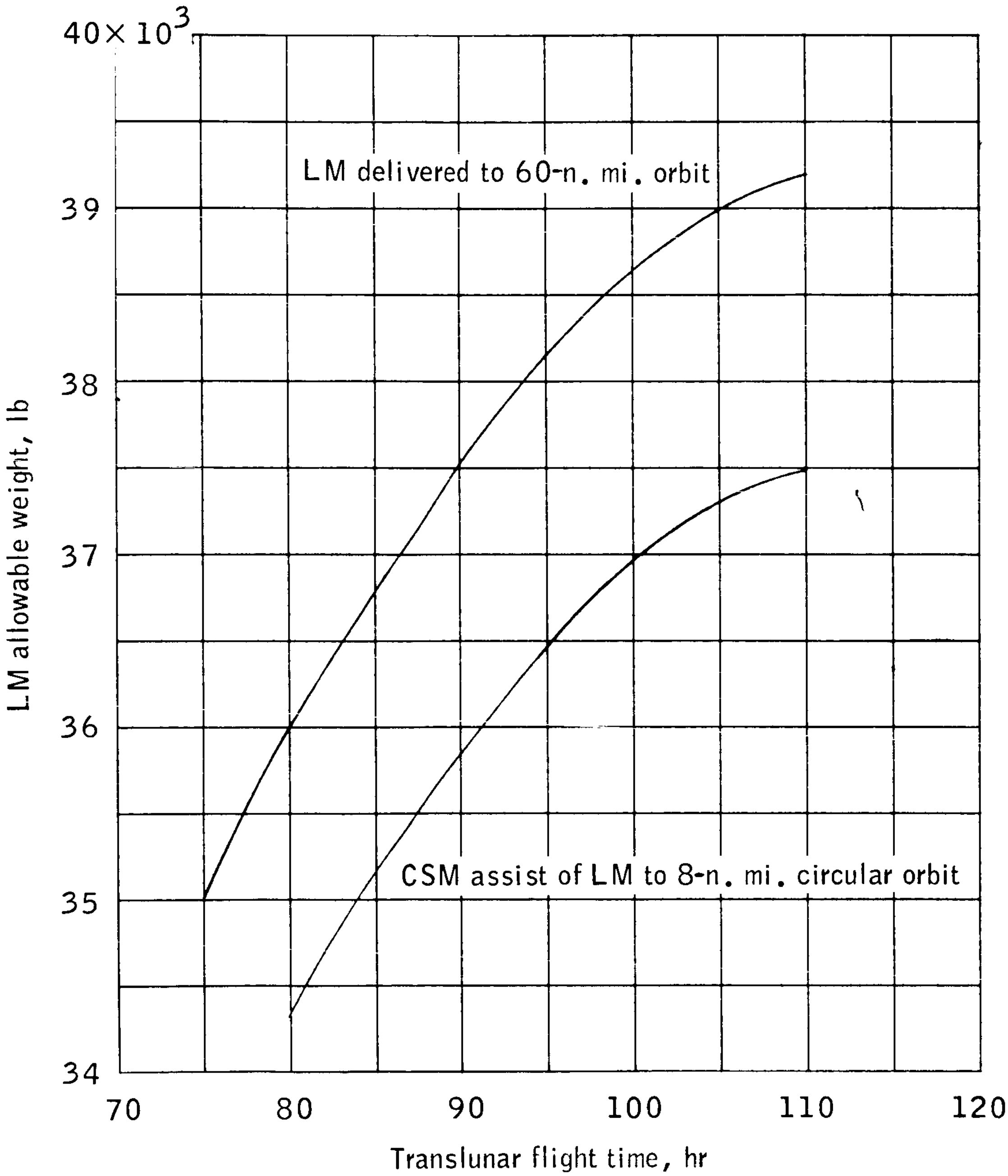


Figure 6.- LM payload as a function of translunar flight time.

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